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INTERFERENCE OF A SWEPTBACK WING AND THE FUSELAGE AT TRANSONIC SPEEDS

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16. Abstract

Calculation results are presented for the interference of a flat sweptback wing and a cylindrical fuselage at transonic speeds and small angles of attack. A random law of spanwise load distribution is used. Results are graphically presented and compared with experimental data.

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ANNOTATION

The results of calculation of interference of a sweptback wing and a cylindrical fuselage at transonic speeds and small angles of attack are presented. A comparison of the data obtained with experimental results demonstrates satisfactory convergence of them.

INTERFERENCE OF A SWEPTBACK WING AND THE FUSELAGE AT TRANSONIC SPEEDS

L. A. Potapova

The problem of flow around a thin, infinite cylinder, with flat, /101
sweptback wings, located in the horizontal plane (Fig. 1), within the framework of theory of a thin body, is reduced to determination of the disturbance velocity potential ϕ, described by the Laplace equation $\phi_{yy} + \phi_{zz} = 0$, under the following boundary conditions, for inductive velocities in the yz plane:

$$\boxed{\begin{array}{lll} V_r = 0, & \rho = r_0, & 0 \leqslant \theta \leqslant 2\pi, \\ V = 0, & y = 0, & t^2 < z^2 < s^2, \\ W = 0, & V = V_0\alpha, & \rho = \infty,\ 0 \leqslant \theta \leqslant 2\pi. \end{array}}$$

Here, V_0 is the velocity of incoming flow; V_r is the velocity normal to the surface of the cylinder; V and W are components of the inductive velocity on the y and z axes, respectively. The meaning of the remaining quantities is clear from Fig. 1.

In work [1], this task is analyzed for the partial case of vortex-free flow around a configuration with the equality $\chi_{pk} = \chi_{zk}$. In this case, it is assumed that there are no free vortexes between the wing and the fuselage. A consequence of this assumption was that the spanwise load was constant at $r^2 < z_1^2 < t^2$, where t = (x) is the distance from the x-axis to the trailing edge of the wing. In this work, the method of determinination of the aerodynamic characteristics of a sweptback wing with a cylindrical fuselage, with an arbitrary principle of spanwise load distribution was used. Circulation in the range $r^2 \leq z^2 \leq t^2$ is presented, in the form of a polynomial of even powers z_1

$$\boxed{\Gamma(z_1) = 2V_0\alpha \sum_{n=0}^{N} b_n z_1^n,} \qquad (1)$$

*Numbers in the margin indicate pagination in the foreign text.

where the expansion coefficients b_n depend on the wing shape in plan and $z_1 = z + \frac{r_0^2}{z}$, $n = 0, 2, 4 \ldots$. In this case, it can be shown that the difference in potentials on the upper and lower surfaces of the wing is determined by the relationship

$$\Delta\varphi = 2V_0\alpha\left\{\frac{s^2 - r_0^2}{s}(E_0 - k_0'^2 F_0) + \sum_{n=0}^{N} b_n\left[\left(s + \frac{r_0^2}{s}\right)^n - \left(t + \frac{r_0^2}{t}\right)^n\right]\right\}, \quad n = 0, 2, 4 \ldots,$$

where $E_0(k_0, \psi_0)$ and $F_0(k_0, \psi_0)$ are incomplete elliptical integrals /102 of the first and second kind, respectively, with the argument

$$\psi_0 = \frac{t}{z}\sqrt{\frac{(s^2 z^2 - r_0^4)(s^2 - z^2)}{(s^2 t^2 - r_0^4)(s^2 - t^2)}}$$

and the modulus

$$k_0 = \frac{\sqrt{(s^2 t - r_0^4)(s^2 - t^2)}}{t(s^2 - r_0^2)},$$

and $k_0'^2 = 1 - k_0^2$.

Factors b_n in expansion (1) depend on the shape of the trailing edge of the wing, and it is found from the following relationship:

$$\frac{s^2 - r_0^2}{s}(E_0 - k_0'^2 K_0) + \sum_{n=0}^{N} b_n\left[\left(s + \frac{r_0^2}{s}\right)^n - 2\left(t + \frac{r_0^2}{t}\right)^n\right] = 0. \qquad (2)$$

For wings with a straight trailing edge $t = t(x)$ a system of linear, nonuniform equations of the N-th order, relative to unknown b_n; $n = 0, 2, 4 \ldots, N$, can be obtained from equality (2). Solution of the system can be obtained on a digital computer. Distribution of the potential differences over the wing surface completely determines the aerodynamic characteristics of the wing, with a fuselage present.

For the configuration represented in Fig. 1, with different values of parameters t_k, r_0, χ_{pk} and χ_{zk}, the total lift and load distribution were determined. The results of the calculations carried out are represented in Figs. 2-5.

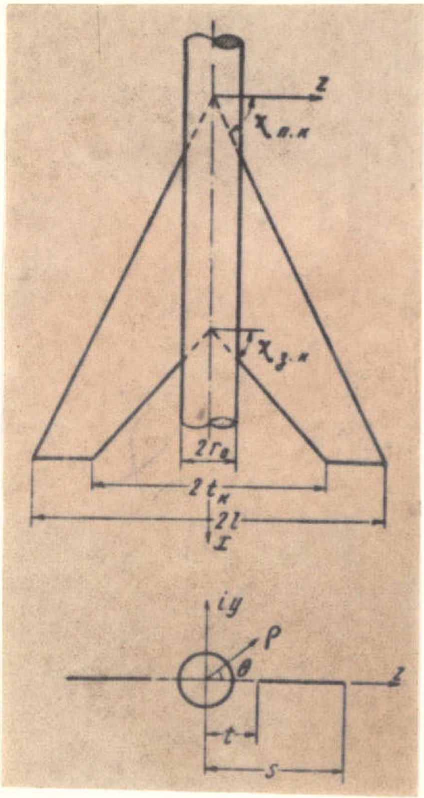

Fig. 1

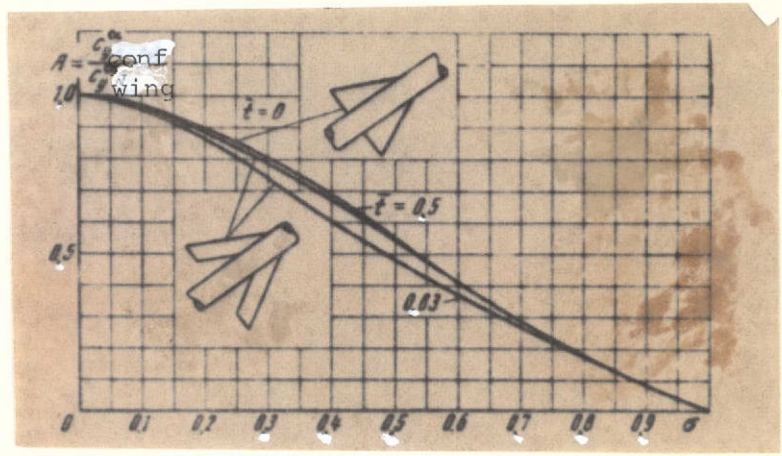

Fig. 2

The dependence of the interference coefficient A, which is the ratio of the lift of the configuration to the lift of a plane sweptback wing having the same shape in plan, to $\sigma = d/2\ell$ and $\bar{t} = t_k/\ell$ is presented in Fig. 2. The calculations show that the interference coefficient for the configuration with a sweptback wing is less than for a configuration with a triangular wing; however, this difference is small. Thus, at $\bar{t} = 0.5$, the interference coefficient for a configuration with the sweptback wing differs by 3.6% from the interference coefficient of the configuration with a triangular wing. Thus, for specific values of $\bar{t}$, this difference can be disregarded.

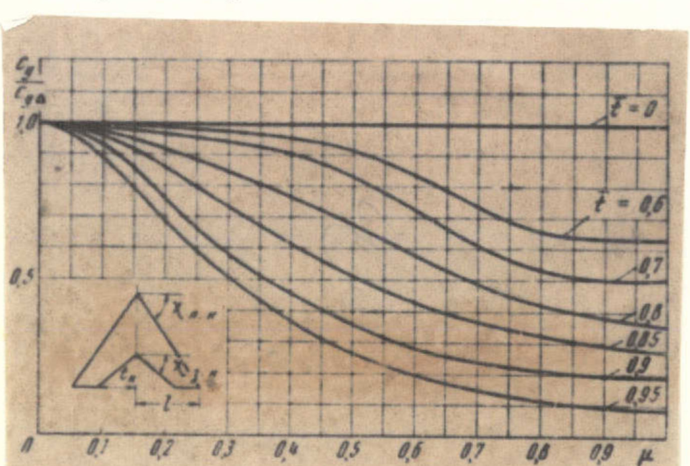

Fig. 3

However, it shoud be kept in mind that the presence of a notch significantly affects the lift of an isolated sweptback wing. As an illustration, $c_y/c_{y\Delta}$ vs. $\mu = \frac{\operatorname{tg}\chi_{з.к}}{\operatorname{tg}\chi_{п.к}}$ (c_y is the lift of an isolated sweptback wing and $c_{y\Delta}$ is the lift of the triangular wing) is given in Fig. 3.

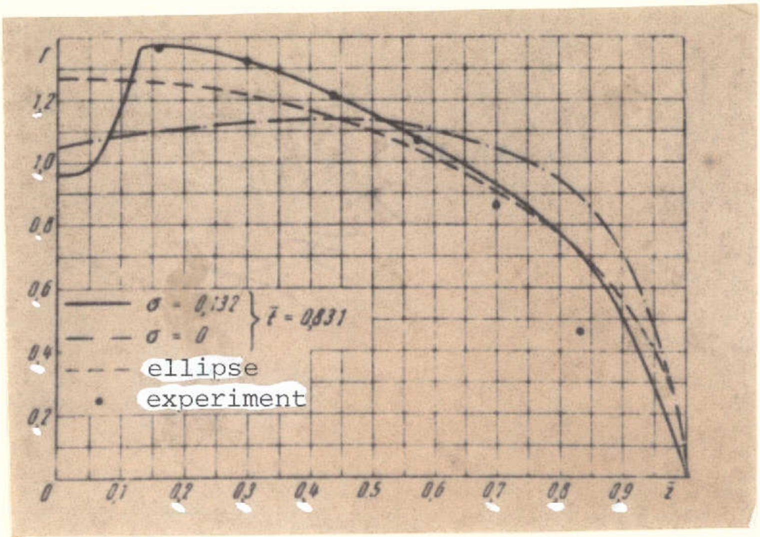

Fig. 4

The distribution of the spanwise circulation Γ for the configuration and for an isolated triangular wing is shown in Fig. 4. The load is redistributed in such a manner, that the cantilever part of the sweptback wing turns out to be in regions of smaller pressure drops than the corresponding part of an isolated triangular wing.

The calculated and experimental data are compared in Figs. 4 and 5. The convergence of the calculated and experimental values of /104
the circulation in the root and central sections (up to z = 0.575) is quite good; however, in sections close to the end of the wing, considerable divergence of them is observed, which is explained by breakaway of the flow.

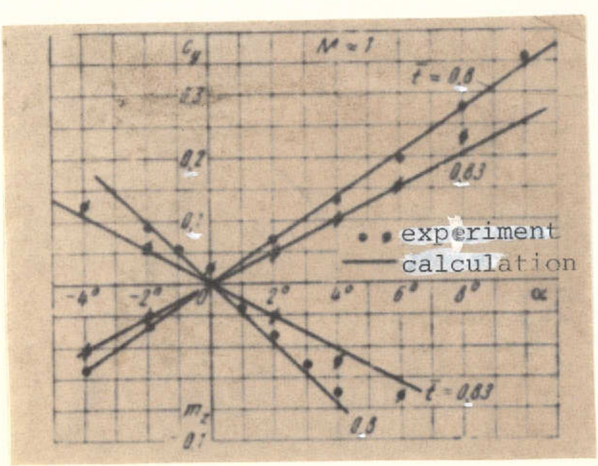

Fig. 5

Functions $c_y(\alpha)$ and $m_z(\alpha)$ are represented in Fig. 5. The calculation results coincide completely satisfactorily with the experimental results, in the linear section of function $c_y = f(\alpha)$. It was assumed in the comparison that the lift of the nose section of the fuselage $c^{\alpha}_{y\ \text{fuse}} = 2\ S_{mid}/S_{wing}$.

In this manner, a comparison of the calculation results and the experimental data shows their satisfactory convergence, in the region of linear dependence of the aerodynamic characteristics on angle of attack.

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